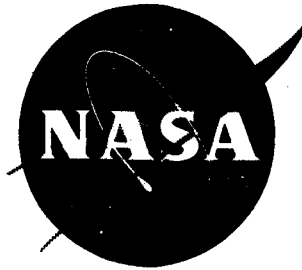


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TECHNICAL NOTE

D-854

INVESTIGATION OF AN AXISYMMETRIC INTERNAL COMPRESSION
INLET AT A MACH NUMBER OF ABOUT 3.8

By John H. Lundell, Richard Scherrer,
and Lewis A. Anderson

Ames Research Center
Moffett Field, Calif.

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INVESTIGATION OF AN AXISYMMETRIC INTERNAL COMPRESSION

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SUMMARY

An idealized axisymmetric, all-internal compression inlet was designed for a Mach number of 3.75. The objective of the design was to obtain a steady, one-dimensional transonic flow and a high over-all total-pressure recovery. Boundary-layer removal was employed in the vicinity of the inflection point of the supersonic contour. Static- and total-pressure fluctuations were measured in the transonic flow region.

A total-pressure recovery of about 90 percent was obtained with a boundary-layer-removal mass-flow rate of 15 percent of the inlet mass-flow rate. The accompanying root-mean-square total-pressure fluctuation in the throat region was only 1 percent of the free-stream total pressure. The test Mach number was 3.80 and the Reynolds number based on inlet diameter was 2.63×10^6 .

INTRODUCTION

A major objective in an inlet design is to obtain a maximum total-pressure recovery which is compatible with other design considerations, such as flow steadiness and distortion, engine matching, over-all drag and weight, and off-design performance. In attempts to attain this objective, early work on supersonic inlets concentrated on the development of the all-external compression type due to its inherent simplicity and therefore low weight. Also, analysis by Rodean (ref. 1) indicates that there is little advantage to be gained using the more complicated all-internal compression inlets for Mach numbers up to 3.0. From Mach number 3.0 to 4.5, however, Rodean's analysis indicates that the all-internal compression inlet has the greater potential. Because of the increased interest in the Mach number range above 3, the all-internal compression inlet has received increased attention.

The results of an investigation of an axisymmetric, all-internal compression inlet, similar to the one of the present report, were reported in reference 2. It was noted in that report that the attainment of high

total-pressure recovery requires a near isentropic supersonic compression followed by a uniform, steady transonic flow with a low terminal shock Mach number. However, such an ideal flow was only partially attained in the inlet of reference 2, and the results indicated the following: (1) the supersonic contour could be improved, (2) the effect of the longitudinal location of the boundary-layer removal scoop should be investigated, and (3) the effect of terminal shock compensation (an abrupt increase in flow area at the throat exit) should also be investigated.

The present investigation was an extension of the work of reference 2, and the purpose was to investigate the three aspects of the inlet design noted above. Four new models were designed and tested at a Mach number of 3.8, Reynolds numbers (based on inlet diameter) ranging from 1.84×10^6 to 2.63×10^6 , and an angle of attack of 0° . During these tests, the total and wall static-pressure fluctuations were measured at several locations in an attempt to relate total-pressure recovery and flow steadiness.

SYMBOLS

A	internal cross-sectional area, sq in.
d_o	inlet diameter, 5.850 in.
l	an arbitrary design length, 16.38 in.
M	Mach number
$\frac{m_{BL}}{m_o}$	ratio of mass-flow rate removed through boundary-layer scoop to inlet capture mass-flow rate
Δp	peak-to-peak pressure difference, inches of mercury
p	static pressure, inches of mercury
p_t	total pressure, inches of mercury
R	Reynolds number based on inlet diameter
r_o	inlet radius, 2.925 in.
r	local internal radius, in.
x	longitudinal length from inlet lip station, in.
y	local distance from surface perpendicular to center line, in.

Subscripts

o	inlet lip station
BL	boundary layer
rms	root mean square
av	area-weighted average at exit station, $\frac{x}{d_o} = 6.95$
∞	free stream

MODEL DESIGN

The models of the present report were designed on the basis of results of the investigation reported in reference 2. These results indicated that the design of the supersonic contour could be improved; they also indicated that the effect of longitudinal location of the boundary-layer-removal scoop and the effect of terminal-shock compensation should be investigated.

The supersonic contour of model 1 (the model of ref. 2) was designed by an approximate analytical method proposed by Foelsch in reference 3. Analysis of the actual flow in this model indicated three major design problems. First, a shock wave occurred at the lip. Second, the Mach waves coalesced too far upstream creating a shock wave with resultant losses about the center line. Last, the longitudinal static-pressure distribution indicated that the supersonic flow was not compressed as much as predicted by the Foelsch method.

In an attempt to avoid the problems associated with model 1, the supersonic contour of the new models was designed by the use of the method of characteristics. The contour from the lip to the inflection point was arbitrarily expressed by a polynomial of the form

$$\frac{r}{r_o} = 1 - K_1 \left[K_2 \left(\frac{x}{l} \right)^a + K_3 \left(\frac{x}{l} \right)^b \right]$$

The coefficients and exponents of the polynomial were varied by trial and error until the theoretical supersonic flow satisfied the following conditions: (1) a zero strength lip Mach wave, (2) coalescence of the initial Mach waves on the wall immediately downstream of the inflection point, and (3) a static-pressure rise (p/p_{∞}) of approximately 9 at the inflection point. The resulting contour was given by the equation

$$\frac{r}{r_0} = 1 - 0.650 \left[0.895 \left(\frac{x}{l} \right)^{2.40} + 0.105 \left(\frac{x}{l} \right)^{4.00} \right]$$

where $r_0 = 2.925$ inches was the radius at the inlet lip and $l = 16.38$ inches was an arbitrary design length. The contour was terminated at $x/d_0 = 2.58$ and $p/p_\infty = 9$, and this was taken as the inflection point of the supersonic contour. In the design of the supersonic contour no attempt was made to account for the boundary-layer growth. The free-stream Mach number used in conjunction with the method of characteristics was 3.75, and that used in conjunction with the Foelsch method was 3.50.

To investigate the effect of the location of the boundary-layer-removal scoop and the effect of terminal-shock compensation, four new models were designed with the supersonic contour described above. The new models are designated 2-F, 2-FC, 2-R, and 2-RC where F and R denote forward and rear boundary-layer-removal scoop locations, respectively, and C denotes the use of terminal-shock compensation. The cross-sectional area distributions of all the models are shown in figure 1. The forward removal scoop was positioned at $x/d_0 = 2.39$ ($p/p_\infty = 5.6$), based on experience gained from the tests of model 1, and the rearward scoop was positioned at the inflection point, $x/d_0 = 2.58$ ($p/p_\infty = 9$). With the removal scoop in the forward position (models 2-F and 2-FC), the supersonic contour from the scoop lip to the inflection point was arbitrarily given by the equation

$$\frac{r}{r_0} = 0.4766 - [0.002002(x-14.00)^{1.463} + 0.07255(x-14.00)]$$

For all four new models, the contour from the inflection point to the throat was arbitrarily given by the polynomial

$$\frac{r}{r_0} = 0.0005908(18.97-x)^{3.64} + 0.3135$$

For models 2-F and 2-R, which had no compensation, the contour from the throat to the subsonic diffuser was faired by a circular arc of radius 3.67 inches. The throat section for models 2-FC and 2-RC, which had compensation, was designed with a 20° included angle conical expansion immediately downstream of the throat. The downstream end of the compensation had a cross-section area 50 percent greater than the throat area (providing a compensation length of 0.6 throat diameter) and was faired into the subsonic diffuser. The diffuser was a simple 4° included angle conical diffuser with an exit area to throat area ratio of 2.33.

The boundary-layer scoop height was designed to remove all the boundary layer and was determined by use of the charts of Simon and Kowalski (ref. 4). A preliminary estimate of the boundary-layer growth indicated that at either removal station the boundary-layer air flow

would comprise approximately 20 percent of the inlet mass flow. An assumed one-fifth power boundary-layer profile was used in conjunction with the charts.

Each model was designed so that the top half of the supersonic contour moved up and forward to reduce the starting contraction ratio and thereby ease the problem of starting supersonic flow in the inlet. Photographs of the inlet in both these positions are shown in figure 2.

WIND TUNNEL, INSTRUMENTATION, AND DATA REDUCTION

Wind Tunnel

All tests were conducted in the 1- by 3-Foot Supersonic Wind Tunnel Number 1 of the Ames Research Center. The tunnel has flexible top and bottom walls which permit variation of the Mach number from subsonic to 6.0. It is a closed system with a stagnation pressure range from 3 to 60 psia.

Instrumentation

Instrumentation was provided for measuring static and total pressures and their fluctuating components in the inlet. Measured pressures include static pressures along the wall, the total pressures across the throat region, the over-all area-weighted total-pressure recovery, and the total pressures across the exit. The static-pressure fluctuations were measured in the transonic region and at the exit, and the total-pressure fluctuation was measured in the transonic region. A schematic diagram of the instrumentation is shown in figure 3.

Instrumentation for static- and total-pressure measurements.- Static and total pressures were measured with liquid-in-glass manometers. The longitudinal static-pressure distribution was obtained by means of static-pressure orifices distributed along the wall. Three static-pressure tubes were mounted in the exit rake. The total-pressure distribution immediately downstream of the terminal shock ($x/d_0 = 3.73$) was measured with a 4-tube rake, and the distribution at exit was measured with a 16-tube rake. The tube spacing in the exit rake was determined by the area-weighted method.

Instrumentation for fluctuating pressure measurements.- A block diagram of the instrumentation used to measure the static-pressure fluctuation on the wall is shown in figure 4. The signal was detected by a transducer and amplified by a 100-kilocycle carrier amplifying system. The voltage fluctuation at the output of the amplifier was recorded on a multichannel tape recorder, which had a flat frequency

response from 0 to 1250 cps, and displayed on a dual channel oscilloscope for visual monitoring. The mean value of the signal was measured with an electronic voltmeter. A variable band-pass filter was adjusted to limit the frequency range displayed on the oscilloscope to the same range recorded on the magnetic tape.

The detector was a capacitance transducer of the conventional deflecting diaphragm type. It was 0.180 inch in diameter, 0.120 inch thick, and incorporated a 0.0005-inch-thick diaphragm. The sensitivity and natural frequency of the transducer were varied by changing the initial tension or the thickness of the diaphragm until the computed natural frequency of the diaphragm was 8100 cps. A description of the transducer and carrier amplifying equipment is given in reference 5, and a typical wall installation of the transducer is shown in figure 4. The cavity in front of the diaphragm was assumed to behave as a Helmholtz resonator, and its natural frequency was computed to be 15,000 cps.

Electrical and pneumatic connections to the transducer were made by means of a stem on the back side of the transducer. The stem was hollow to permit a reference pressure or a calibration pressure to be applied to the back side of the diaphragm. In all cases the reference pressure was detected by means of a static-pressure orifice or a total-pressure tube in the immediate vicinity of the transducer. All fluctuations in the reference pressure were damped out by means of a 10-foot length of 0.021-inch I.D. flexible tubing. In essence this procedure removed the bias from the signal, leaving only the fluctuating component.

A static calibration of the transducer-amplifier combination was accomplished by applying a known pressure difference across the transducer and measuring the output signal with an electronic voltmeter. Such a calibration was made before and after each test, and a typical example is shown in figure 5. Since the bias was removed, the signal fluctuated about a neutral position which allowed maximum use of the linear range of the curve.

Except for a total-pressure tube, the instrumentation for measuring the total-pressure fluctuation was identical to that described above. A schematic diagram of the tube, which was mounted adjacent to one of the four conventional tubes in the throat rake, is shown in figure 6. Such a tube must be designed in a manner such that the signal being recorded is essentially the same as the signal being detected over the required frequency range. The frequency response of the total-pressure tube and transducer combination was determined by means of a resonant "organ pipe" with a movable piston. A schematic diagram of the test equipment is shown in figure 7. The open end of the pipe was excited by a speaker vibrator which was driven by the amplified output of an audio oscillator. A simulated total-pressure tube with a transducer at one end was mounted with its open end flush with the face of the piston. In order to detect the input signal, a reference transducer was also mounted flush with the

face of the piston and diametrically opposite the total-pressure tube. The sensitivities of both transducer-amplifier combinations were adjusted until equal, and the two sinusoidal signals were displayed on a dual trace oscilloscope for a direct comparison of their amplitudes. The phase relationship between the two signals was obtained from a Lissajous figure which was displayed on an x-y plotting oscilloscope. Initial calibration of the combination of transducer and the total-pressure tube indicated that the system was very lightly damped. An increase in the damping was achieved by means of a 0.010-inch I.D. by 0.625-inch-long tube inserted in the open end of the total-pressure tube. The final amplitude and phase calibrations of the total-pressure tube are shown in figure 8.

Data Reduction

The over-all total-pressure recovery was computed from the exit rake pressures by the area-weighted method discussed in reference 6. The bleed flow rates were computed as the difference between the inlet mass-flow rate and the exit mass-flow rate which was determined from the exit rake.

The pressure fluctuation measurements were reduced in terms of a peak-to-peak unsteadiness parameter and a root-mean-square unsteadiness parameter. The peak-to-peak parameter is defined as the maximum pressure minus the minimum pressure divided by the free-stream total pressure; and the root-mean-square parameter is defined as the ratio of the rms pressure to the free-stream total pressure. In general, the wave form of any of the signals was random, and in order to obtain a representative sample, the data were recorded for a period of at least 30 seconds for each data point. For the purpose of data analysis, the recorded signal was played into an rms meter or displayed by an oscilloscope.

The peak-to-peak values were determined by photographing the display on the oscilloscope. The sweep rate of the oscilloscope was adjusted to 10 seconds for a complete sweep, and photographs of three successive sweeps were taken for each data point. In general, the measured peak-to-peak values were not the same for all three sweeps. In the worst case, however, the difference between the maximum and the minimum values of the peak-to-peak parameter for a given data point was 18 percent, and in most cases it was 10 percent or less.

The root-mean-square value of each random signal was obtained with a Ballantine true rms electronic voltmeter. The meter has a specified accuracy of 5 percent for signals with a frequency content between 5 and 50,000 cps.

To illustrate the random but axially symmetric nature of the flow, photographs of the "raw" signals from diametrically opposite transducers

were obtained from the display on the dual trace oscilloscope. The data were played off the magnetic tape into the oscilloscope in which the single beam was chopped at a rate of 100,000 times per second. The high-speed switching produced two virtually simultaneous traces.

RESULTS

The total-pressure recovery and bleed flow fraction are summarized in table I. The longitudinal static-pressure distributions to the inflection point of the supersonic contour of the present models and the model of reference 2 are shown in figure 9 to provide a comparison of the results of the two design methods. The effect of the location of the boundary-layer-removal scoop and the effect of the use of terminal-shock compensation on the longitudinal static-pressure distribution of the present models are shown in figure 10; total-pressure boundary-layer profiles are presented in figures 11 and 12.

The pressure fluctuation data are summarized in table II; a relationship between total-pressure recovery and flow steadiness is presented in figure 13; and oscilloscope pictures of static- and total-pressure fluctuations are shown in figure 14. Except as noted, all data are for the maximum value of total-pressure recovery.

DISCUSSION

Static- and Total-Pressure Data

The effect of boundary-layer-bleed scoop location and terminal-shock compensation on the over-all total-pressure recovery can be ascertained from the data summarized in table I. The corresponding free-stream Mach number, Reynolds number, and measured boundary-layer bleed fraction are included. The highest value of total-pressure recovery (89.8 percent) was attained by model 2-R which had the boundary-layer-removal scoop in the rear position and no terminal-shock compensation.

A comparison of the theoretical and experimental longitudinal static-pressure distributions of the present models with those of model 1 is given in figure 9. In the case of the more exact design method (present models), the experimental compression exceeds the theoretical compression. This difference can be accounted for by the fact that the theoretical method does not account for boundary-layer growth. In the case of the approximate design method (model 1), the experimental compression is less than the theoretical compression, indicating that the method is inadequate for supersonic diffuser design. Data obtained but not shown indicate that in either case the difference in Mach number had an insignificant effect on the pressure distributions.

The importance of the location of the boundary-layer-removal scoop and the effect of terminal-shock compensation can be ascertained from the longitudinal static-pressure distributions shown in figure 10. It is evident that at the exit of the subsonic diffuser the static-pressure ratio is higher for the rear position of the scoop than for the forward position. The effect of terminal-shock compensation is to reduce the static-pressure ratio and the over-all total-pressure recovery. Thus, if any benefits are to be attained from the use of terminal-shock compensation, further experimental evaluation is necessary.

The total-pressure profiles downstream of the terminal-shock wave (fig. 11) show that the total-pressure losses in the boundary layer are smallest for the scoop in the rear position and for no terminal-shock compensation. It can be seen from the curve for model 2-FC that the total pressure is approximately constant for some distance from the wall. If it is assumed that the static pressure is also constant, separated flow is indicated for this model. Model 2-R, which had the highest over-all total-pressure recovery, had the smallest losses and no apparent separation at this station.

The distortion of the exit flow is shown by the total-pressure profiles of figure 12. The radial distortion, which is defined as the ratio of the difference between the maximum and minimum total pressures to the average total pressure, is 5.7 percent or less for all the models. The circumferential distortion, which is defined as the ratio of the difference between the maximum and minimum total pressures at a given radial station to the average total-pressure, is 1.5 percent or less. The position of the boundary-layer-removal scoop had little effect on the distortion. Terminal-shock compensation, however, reduced the distortion somewhat.

Pressure Fluctuation Data

To evaluate the effect of pressure fluctuation on inlet performance, the peak-to-peak and root-mean-square static- and total-pressure unsteadiness parameters were computed and are summarized in table II. For comparison the corresponding Reynolds number and total-pressure recovery are included. (Due to a malfunction in the electronic apparatus, the signal from transducer number 1 was not recorded during the test of model 2-FC.) The relationship between total-pressure recovery and root-mean-square flow unsteadiness is best illustrated by the total- rather than the static-pressure unsteadiness. In all cases shown in the table, the total-pressure unsteadiness is larger (in some cases by an order of magnitude) than the static-pressure unsteadiness. This same result is evident in the time-dependent boundary-layer profiles presented in reference 2 and indicates the need for measuring total- as well as static-pressure unsteadiness.

It is evident from the data presented in table II that the removal of terminal-shock compensation and the placement of the boundary-layer scoop in the rear position had the same effect on flow steadiness as they did on total-pressure recovery. Thus the two changes in geometry decreased the flow unsteadiness on virtually all traces. Removal of the terminal-shock compensation had the largest effect. It should be noted that model 2-R had both the highest total-pressure recovery (89.8 percent) and the lowest total-pressure unsteadiness (1 percent).

The relationship between total-pressure recovery and total-pressure unsteadiness is illustrated in figure 13. The increase in total-pressure recovery with decreasing total-pressure unsteadiness is in qualitative agreement with the work of Kantrowitz (ref. 7). In this early work on the stability of transonic flows, it was concluded that smooth transonic flow was unstable to compression pulses coming from downstream. It was further hypothesized that the disturbance level could affect the minimum shock strength which could be attained in a supersonic diffuser. Thus, to the extent that it depends on terminal-shock strength, the over-all total-pressure recovery depends on the unsteadiness of the flow. In addition, unsteady flow can adversely affect the pressure recovery through time-dependent separation and distortion.

To illustrate the random nature of the pressure fluctuations in the throat region, photographs of the "raw" signals are reproduced in figure 14. Traces 1 and 2, and traces 3 and 4 are grouped together since the corresponding transducers are at the same longitudinal station in the models. The vertical gain of the oscilloscope was adjusted so that the sensitivity for traces 1, 2, 3, and 4 in inches of mercury per division of the grid is the same for a given data point. In most cases it was necessary to reduce the sensitivity of the total-pressure trace (number 5) to a fraction of the sensitivity of the other traces. The "raw" signals show that the pressure fluctuations are random and that the low frequency correlation between diametrically opposite traces 1 and 2 is very good. Thus, in the throat region the terminal-shock motion is planar.

The static-pressure fluctuation at the exit of the subsonic diffuser (fig. 14(e)) was measured for one operating condition of model 2-F. It should be noted that this was not the maximum pressure recovery condition for this model. The exit station data, which are listed as trace 6 in table II, indicate that the exit flow is very steady.

CONCLUDING REMARKS

Tests were conducted on an idealized axisymmetric, all-internal compression inlet at a Mach number of 3.8 and Reynolds numbers (based on inlet diameter) of 1.84×10^6 and 2.63×10^6 . Boundary-layer removal was

investigated at a forward and at a rearward location in the vicinity of the inflection point of the supersonic contour. The inlet was tested with and without terminal-shock compensation.

The model with the rear boundary-layer-removal position and without terminal-shock compensation attained the highest total-pressure recovery. At its best operating condition, with 15 percent of the entering mass-flow rate being removed through the boundary-layer scoop, the total-pressure recovery was 89.8 percent, and the total-pressure unsteadiness downstream of the terminal shock was 1.0 percent of the free-stream total pressure.

In general, the total-pressure recovery increased with decreasing flow unsteadiness. The model with the most steady transonic flow had a planar terminal shock and the highest total-pressure recovery. The total-pressure unsteadiness was always larger than the static-pressure unsteadiness at the wall.

Ames Research Center

National Aeronautics and Space Administration

Moffett Field, Calif., April 11, 1961

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TABLE I.- SUMMARY OF OVER-ALL PERFORMANCE

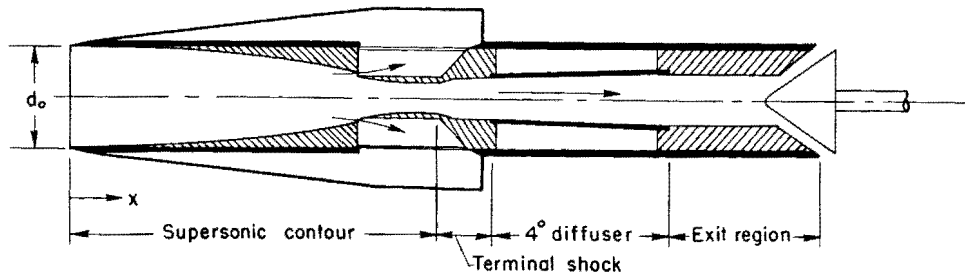
Model designation	Location of removal scoop, x/d_o	Terminal-shock compensation	M_{to}	$R \times 10^{-6}$	$\frac{m_{BL}}{m_o}$	$\left(\frac{p_t}{p_{t_{\infty}}}\right)_{av}$
1	2.84	yes	3.70	1.8	0.20	0.848
2-FC	2.39	yes	3.82	2.6	.15	.814
2-F	2.39	no	3.81	2.6	.15	.864
2-RC	2.58	yes	3.80	2.6	.15	.863
2-R	2.58	no	3.80	2.6	.15	.898

TABLE II.- FLUCTUATING PRESSURE DATA

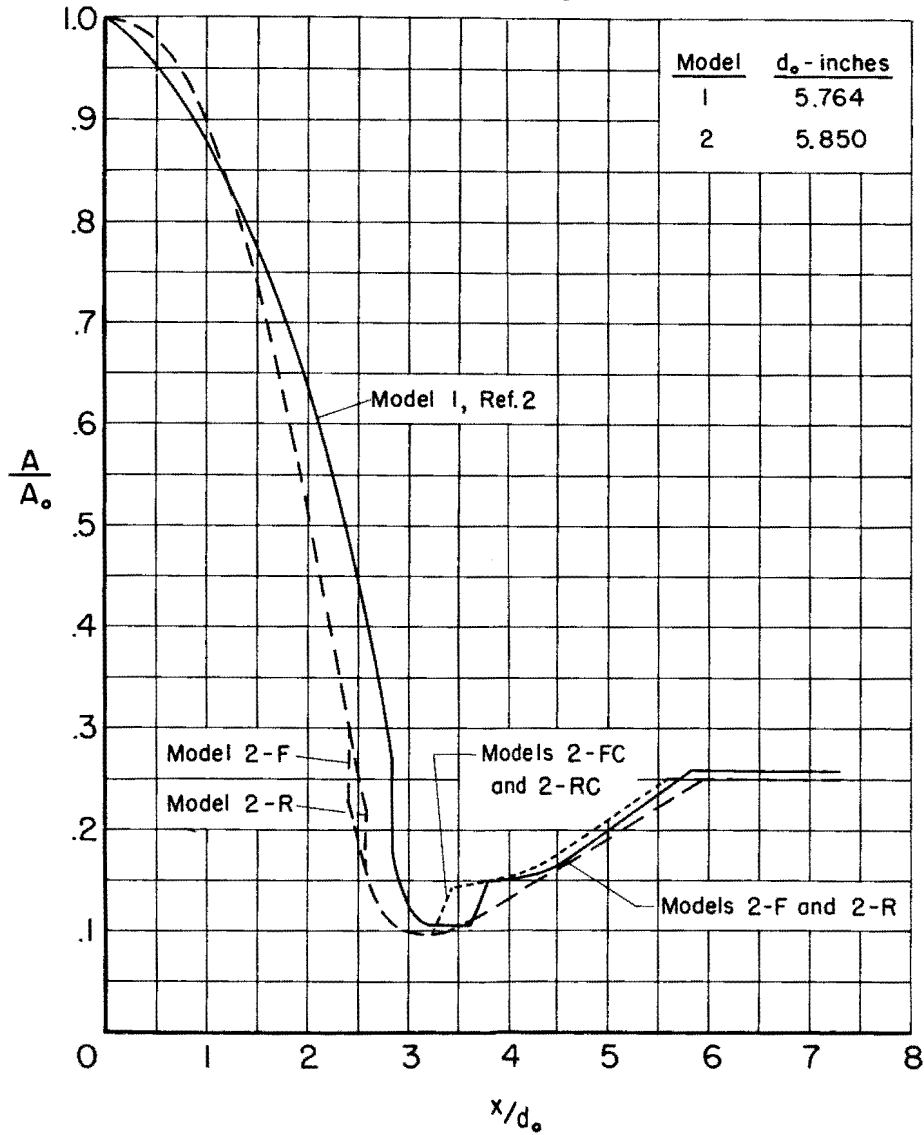
Model designation	Reynolds number $\times 10^{-6}$	Pressure recovery $\left(\frac{p_t}{p_{t\infty}}\right)_{av}$	Peak-to-peak unsteadiness, percent					
			$x/d_o = 3.17$		$x/d_o = 3.73$			$x/d_o = 6.74$
			Trace 1 $\Delta p/p_{t\infty}$	Trace 2 $\Delta p/p_{t\infty}$	Trace 3 $\Delta p/p_{t\infty}$	Trace 4 $\Delta p/p_{t\infty}$	Trace 5 $\Delta p_t/p_{t\infty}$	Trace 6 $\Delta p/p_{t\infty}$
2-FC	1.84	0.808		3.08	6.64	6.55	32.4	2.04
2-FC	2.63	.814		7.38	4.80	8.63	40.9	
2-F	1.84	.805	2.15	2.75	3.01	3.20	22.0	
2-F	2.63	.831						
2-F	2.63	.864	5.64	5.60	1.95	1.94	15.9	
2-RC	1.84	.849	8.71	10.90	9.53	8.68	33.3	
2-RC	2.63	.863	6.32	7.58	6.60	7.06	22.9	
2-R	2.63	.898	2.56	2.22	1.10	1.23	6.02	

Model designation	Reynolds number $\times 10^{-6}$	Pressure recovery $\left(\frac{p_t}{p_{t\infty}}\right)_{av}$	Root-mean-square unsteadiness, percent					
			$x/d_o = 3.17$		$x/d_o = 3.73$			$x/d_o = 6.74$
			Trace 1 $p_{rms}/p_{t\infty}$	Trace 2 $p_{rms}/p_{t\infty}$	Trace 3 $p_{rms}/p_{t\infty}$	Trace 4 $p_{rms}/p_{t\infty}$	Trace 5 $p_{t_{rms}}/p_{t\infty}$	Trace 6 $p_{rms}/p_{t\infty}$
2-FC	1.84	0.808		0.172	0.736	0.763	4.71	0.20
2-FC	2.63	.814		1.56	.548	.933	6.90	
2-F	1.84	.805	0.241	.297	.359	.359	3.88	
2-F	2.63	.831						
2-F	2.63	.864	.879	.899	.232	.218	2.02	
2-RC	1.84	.849	1.57	1.66	.472	.566	4.67	
2-RC	2.63	.863	1.16	1.14	.727	.905	3.85	
2-R	2.63	.898	.351	.344	.131	.137	1.02	

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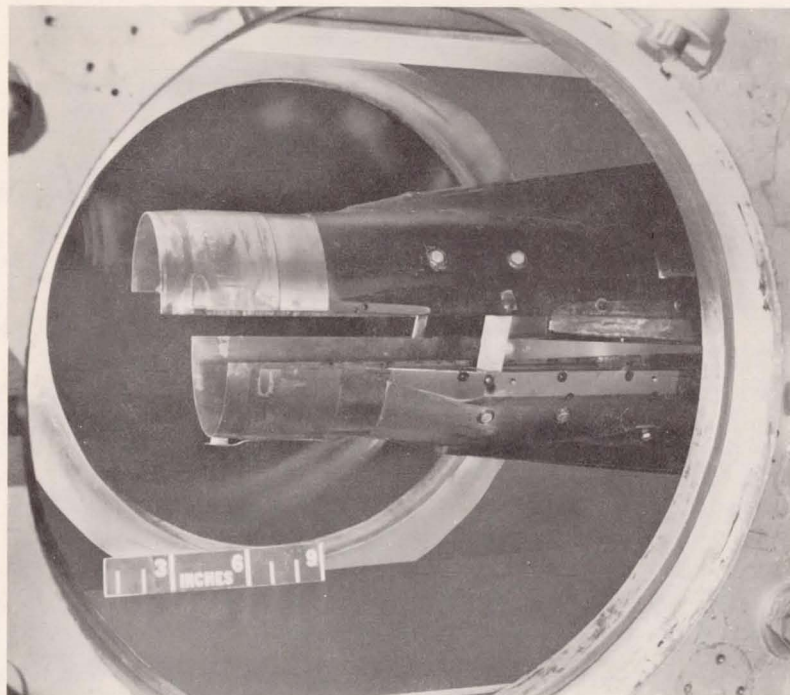


(a) Schematic drawing of models.



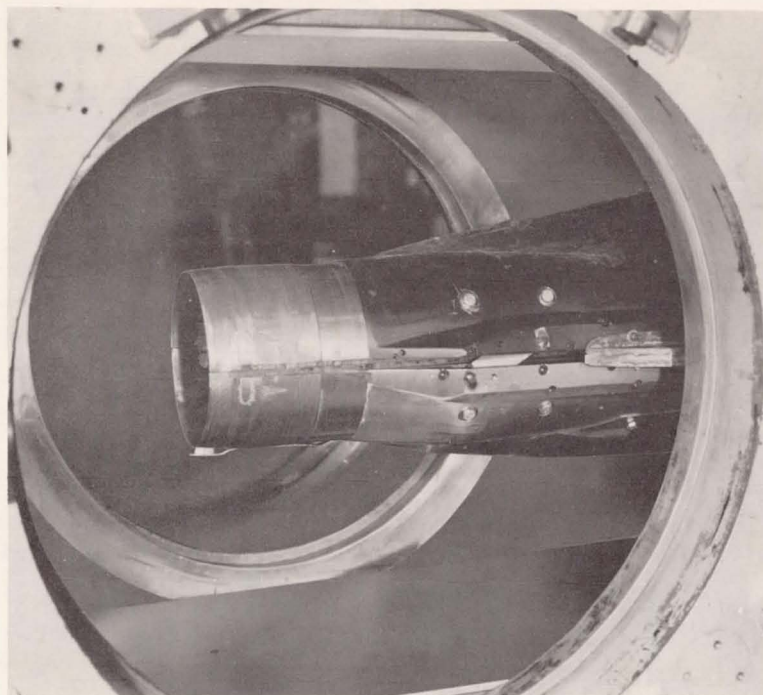
(b) Longitudinal area distributions.

Figure 1.- Schematic drawing and longitudinal area distributions of models.



Model with top half in
position for starting

A-23547.1



Model in started position

A-23548.1

Figure 2.- Internal compression inlet mounted in the wind tunnel.

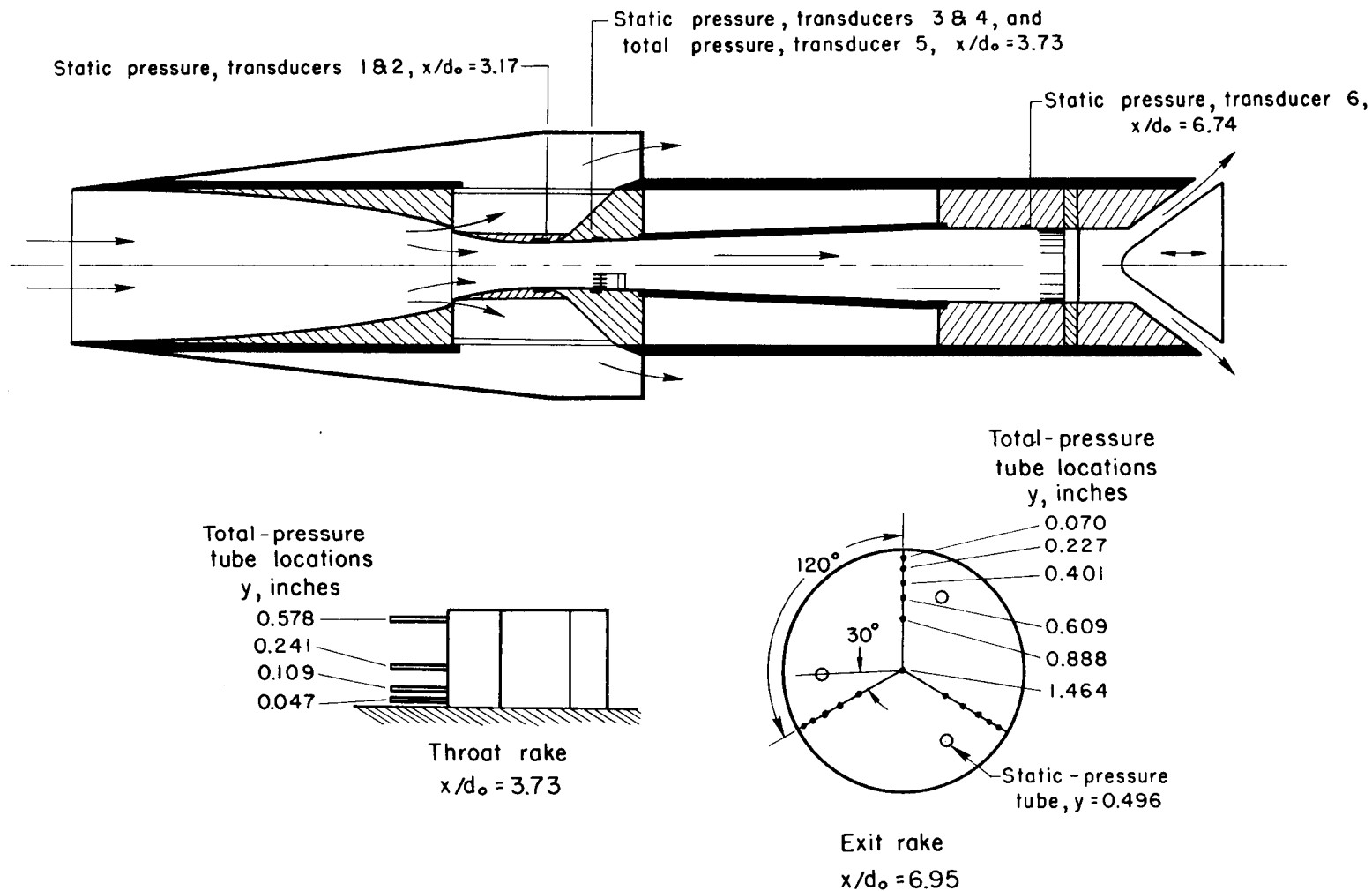


Figure 3.- Location of instrumentation.

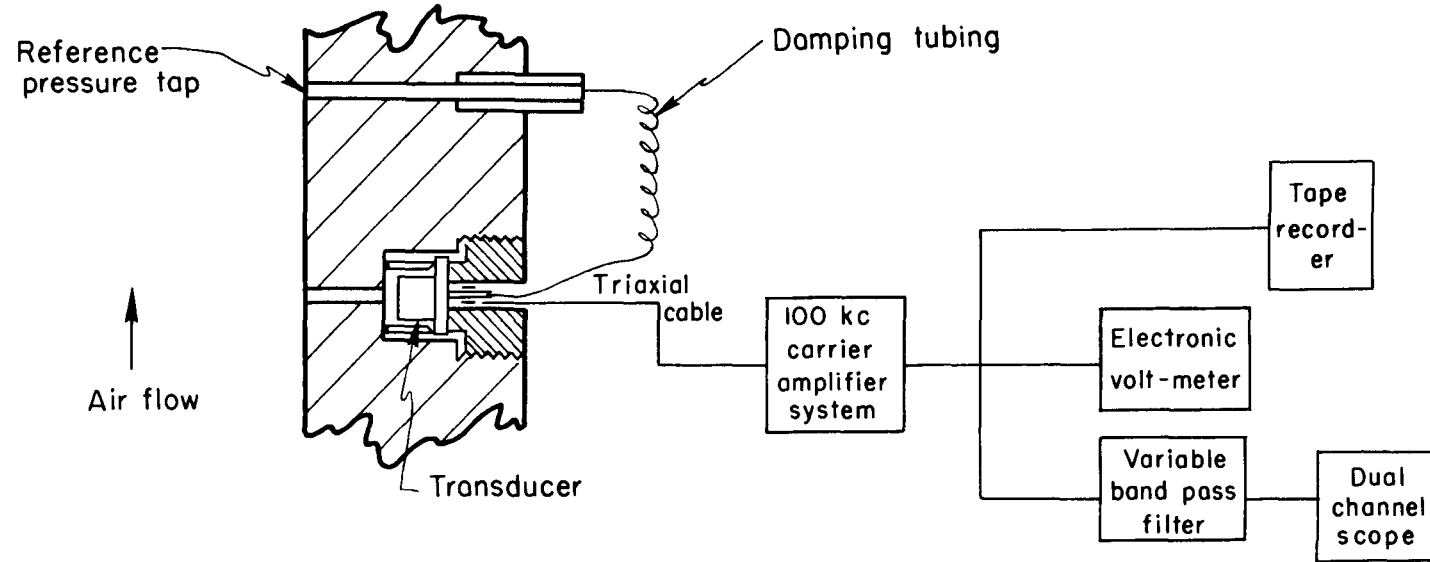


Figure 4.- Equipment for measuring pressure fluctuations.

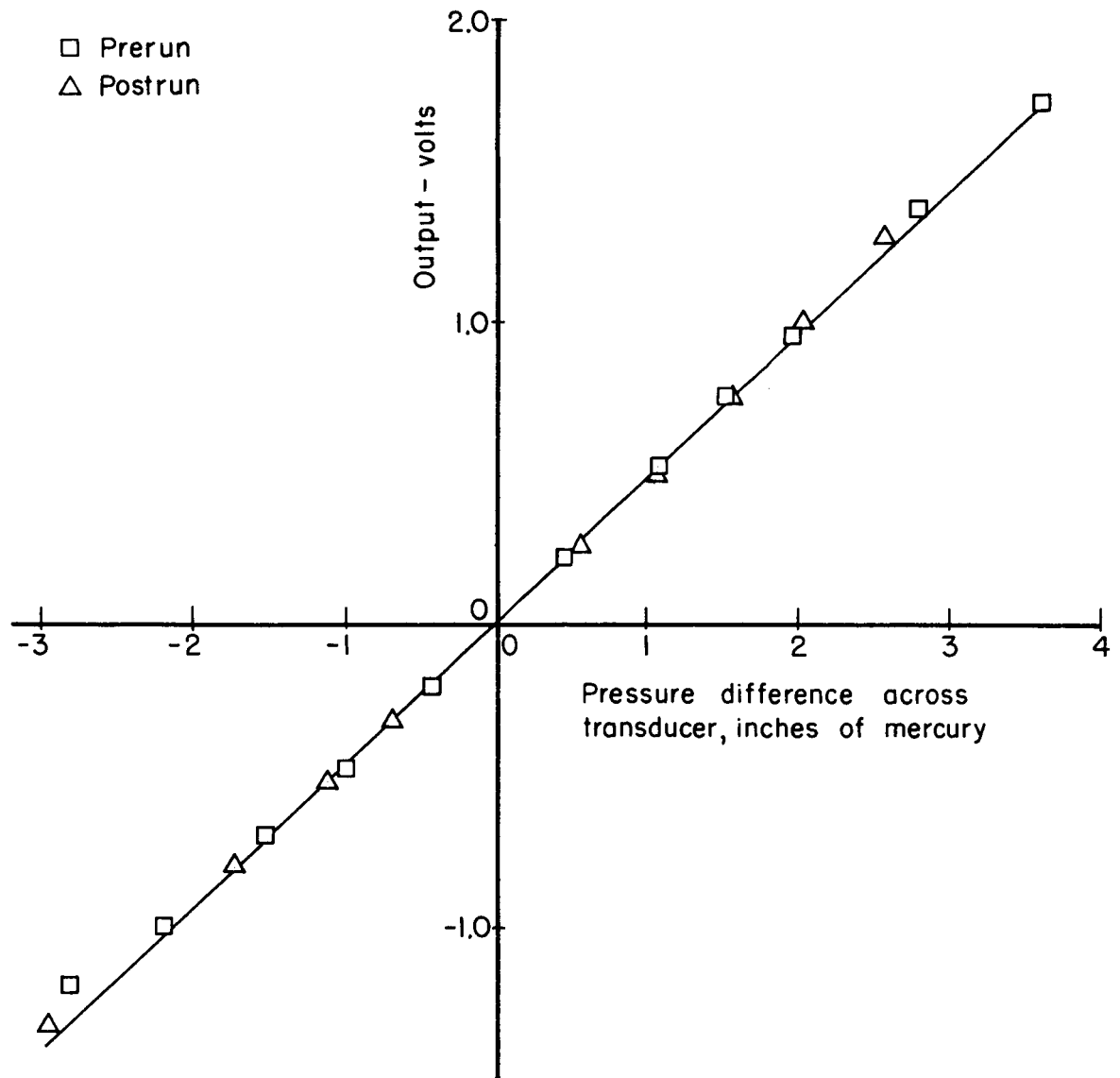


Figure 5.- Typical calibration curve for transducer.

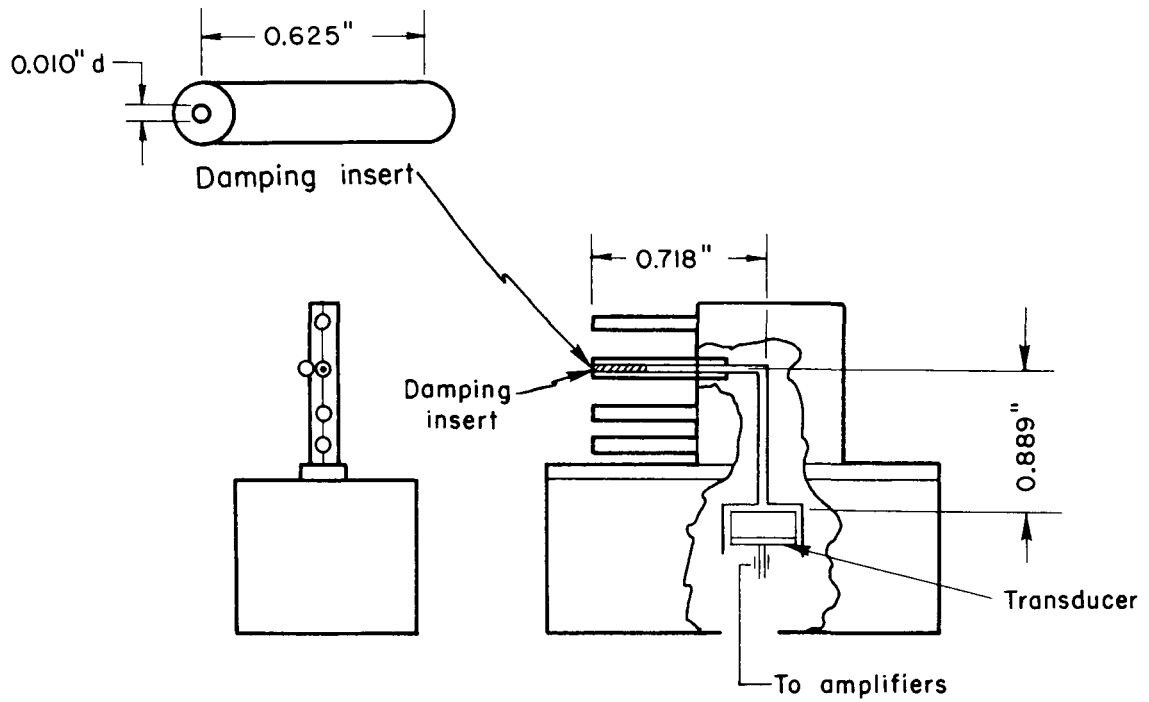


Figure 6.- The high-frequency total-pressure tube and throat rake.

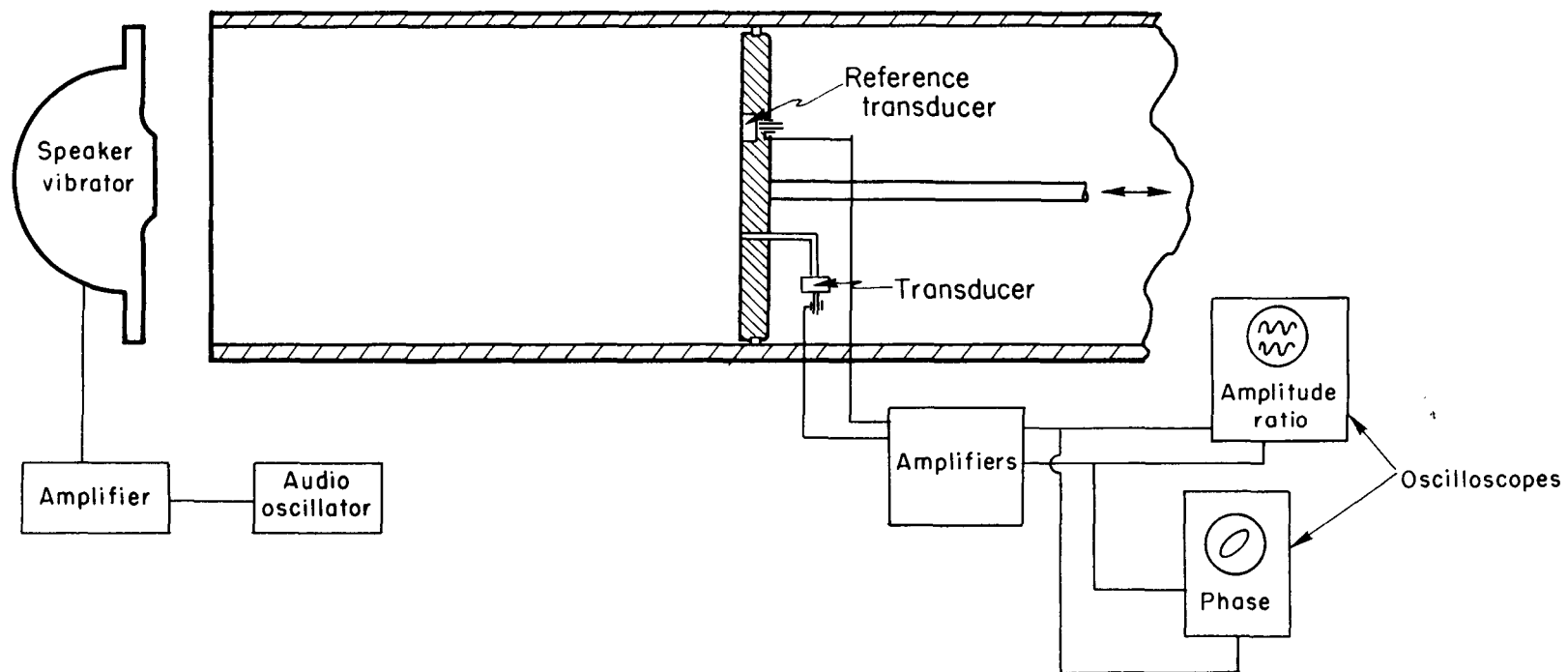


Figure 7.- Calibration equipment for the high-frequency total-pressure tube.

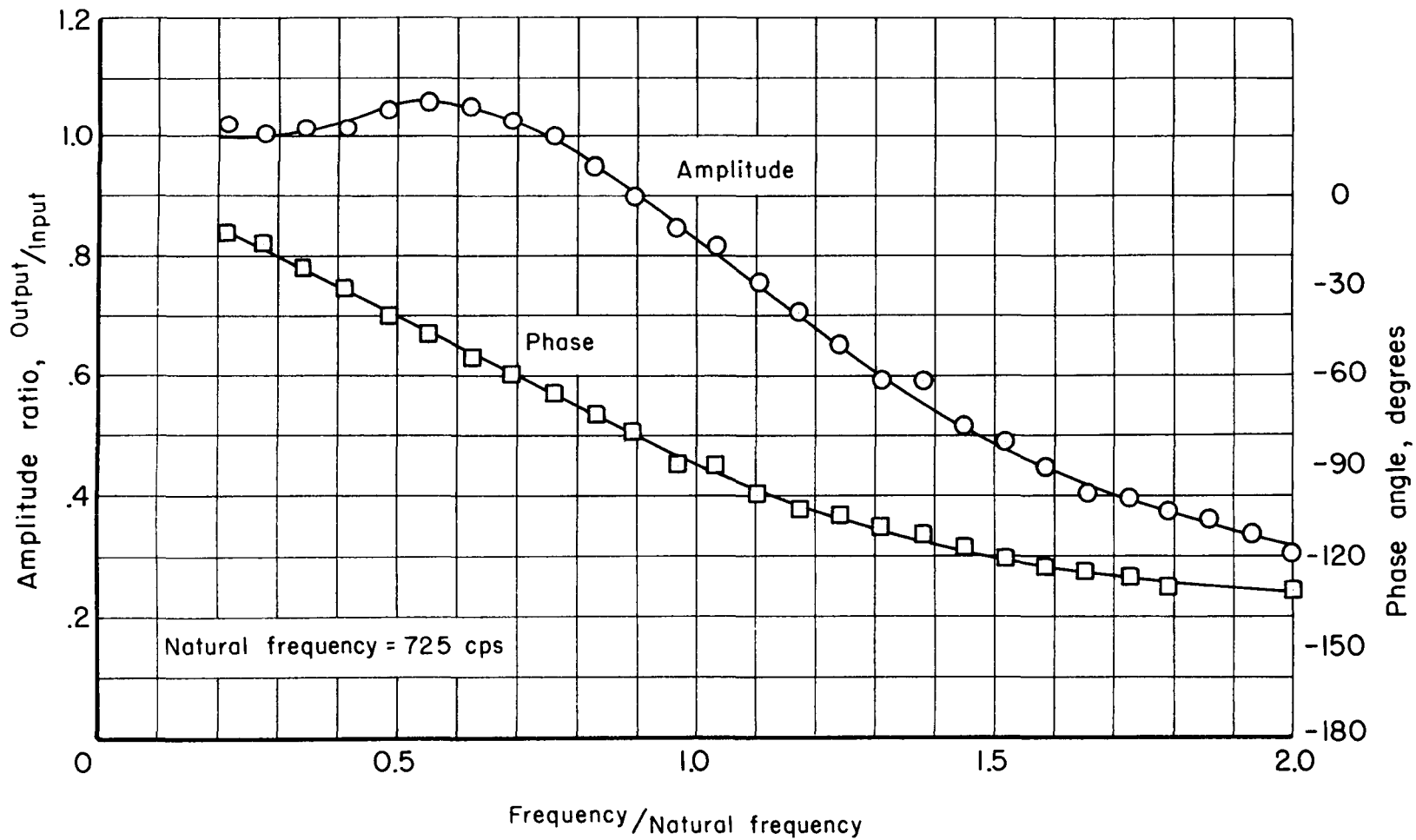


Figure 8.- Frequency response of total-pressure tube.

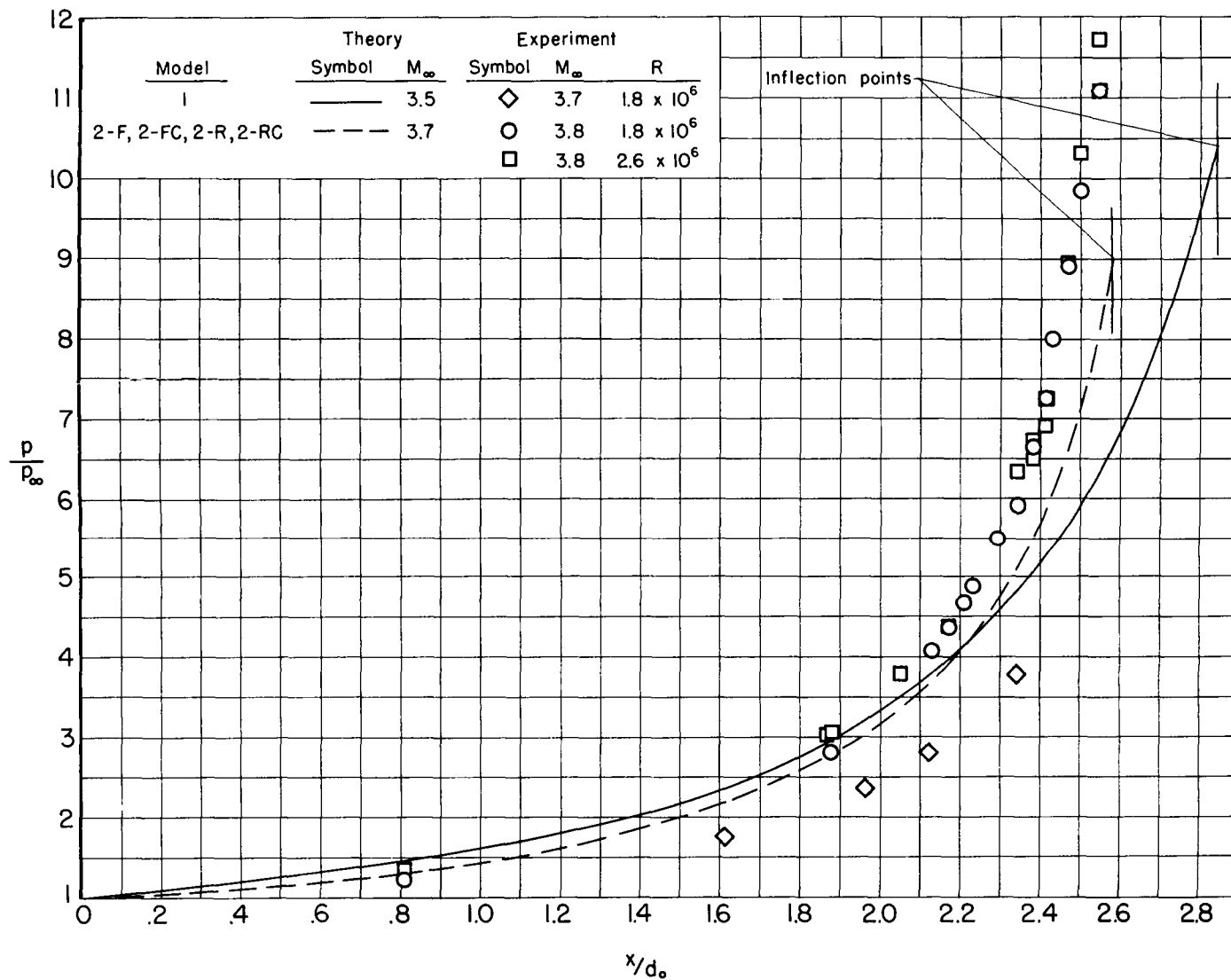


Figure 9.- Theoretical and experimental static-pressure distributions in the supersonic contour upstream of the inflection point.

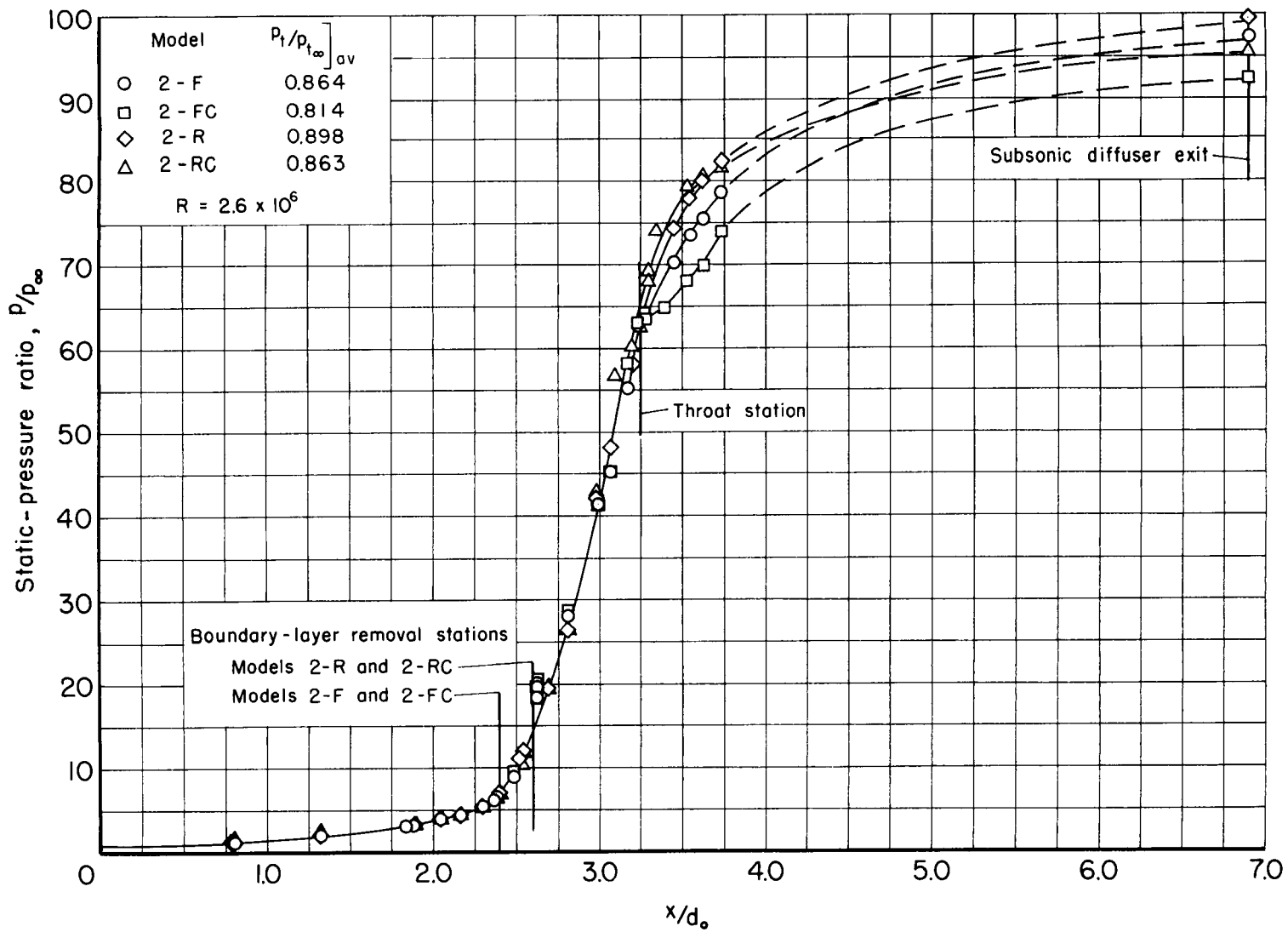


Figure 10.- Experimental longitudinal static-pressure distribution for new models.

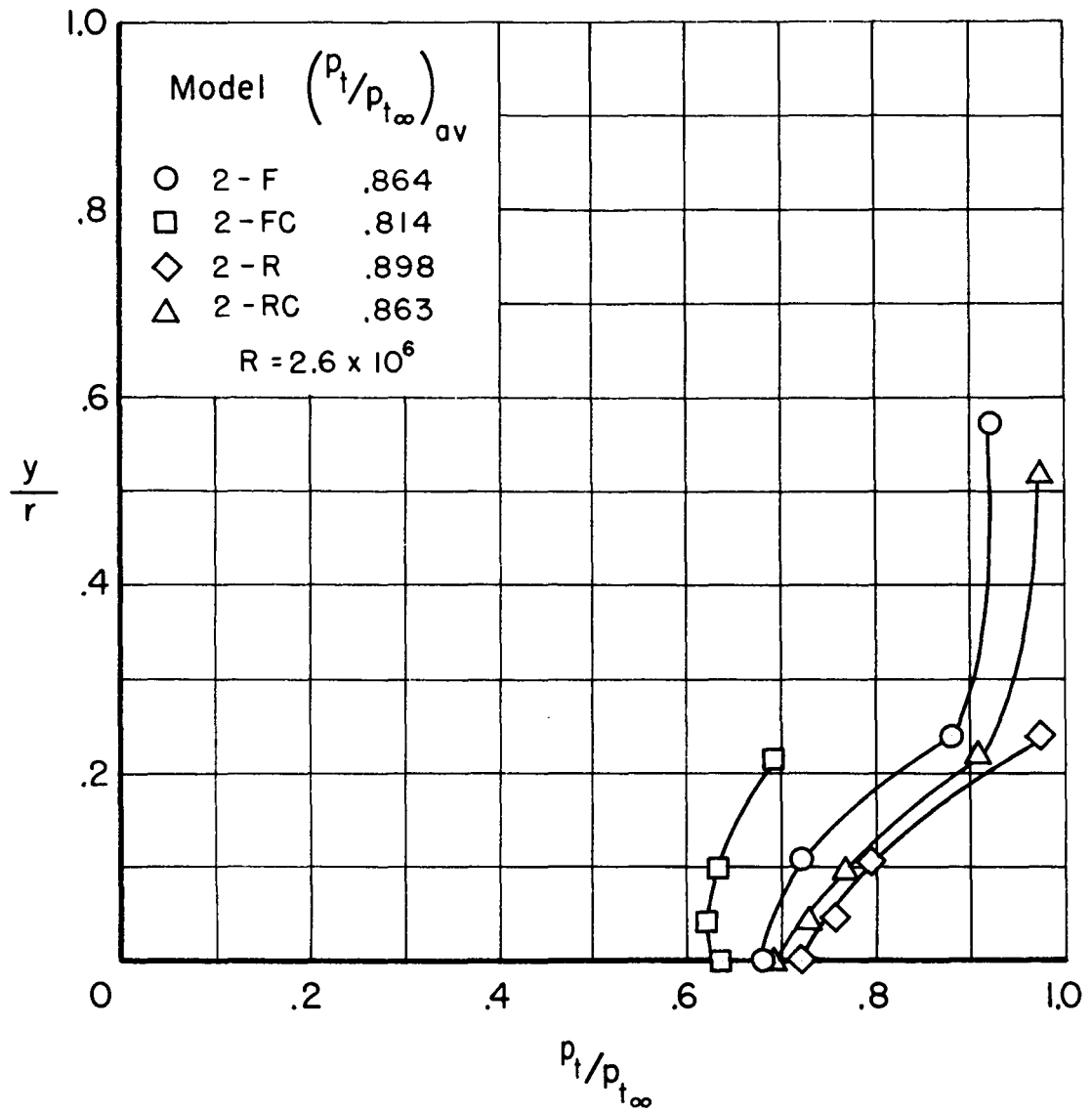
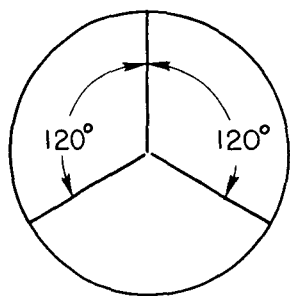


Figure 11.- Total-pressure boundary-layer profiles downstream of the terminal shock; $x/d_0 = 3.73$.



Model	$(p_t/p_{t\infty})_{av}$	Radial distort. $\Delta p_t/p_{t_{av}}, (\%)$	Cir. distort. $\Delta p_t/p_{t_{av}}, (\%)$
○ 2-F	0.864	5.7	1.5
□ 2-FC	0.814	1.9	0.8
◇ 2-R	0.898	5.6	0.7
△ 2-RC	0.863	3.7	0.9

$$R = 2.6 \times 10^6$$

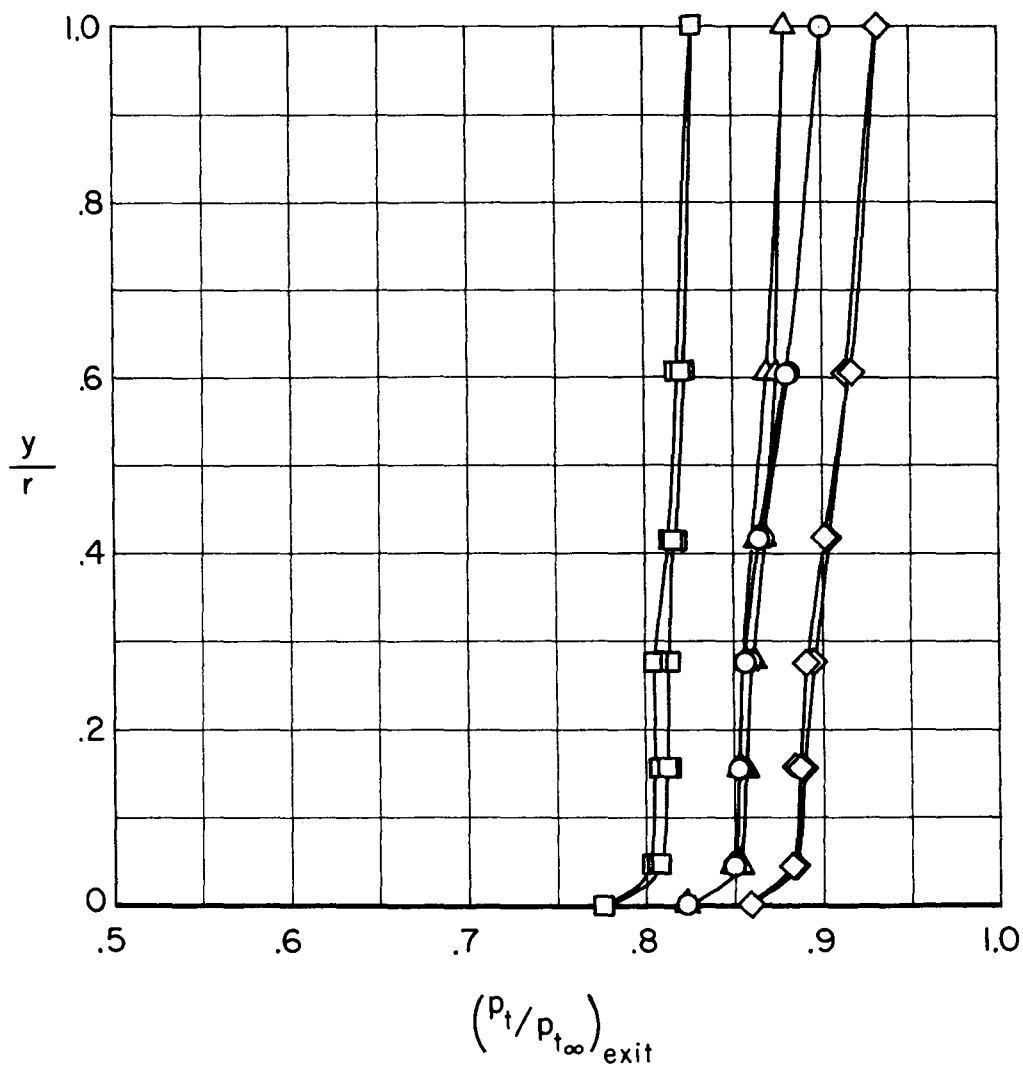


Figure 12.- Total-pressure profiles at the exit of subsonic diffuser;
 $x/d_o = 6.95$.

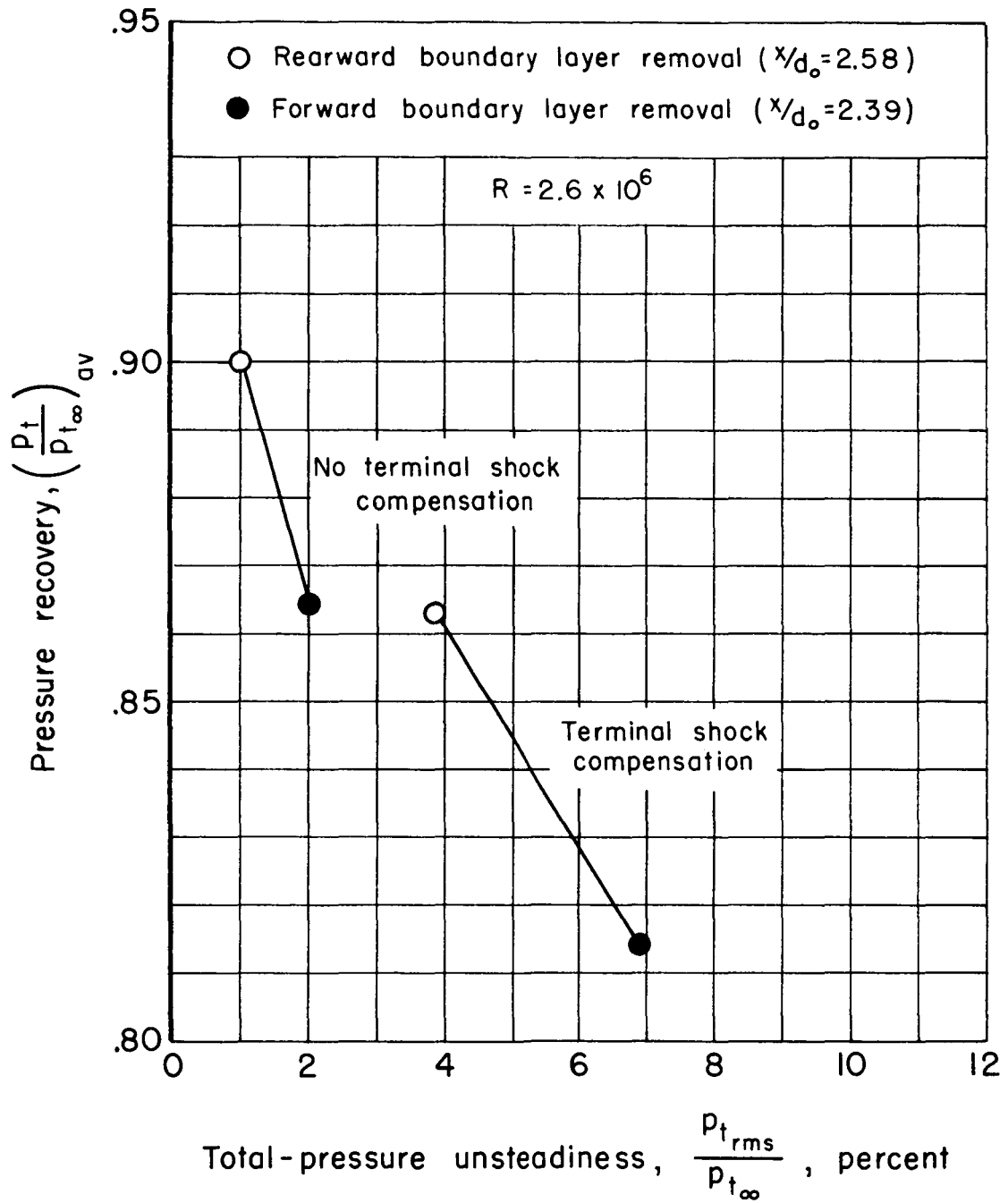
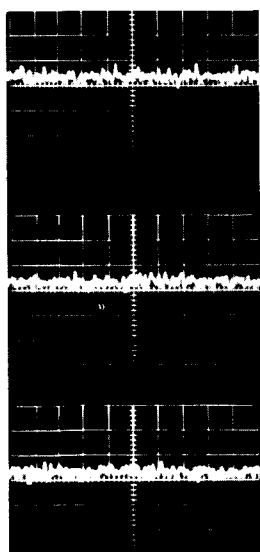
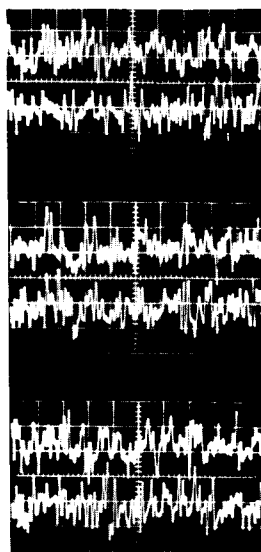


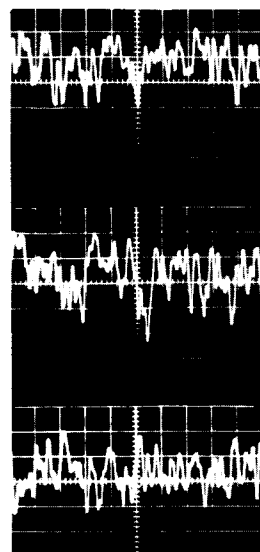
Figure 13.- Pressure recovery as a function of total-pressure unsteadiness.



Static pressure
Trace 2, $x/d_o = 3.17$
Gain = 1.06 in. Hg/div.

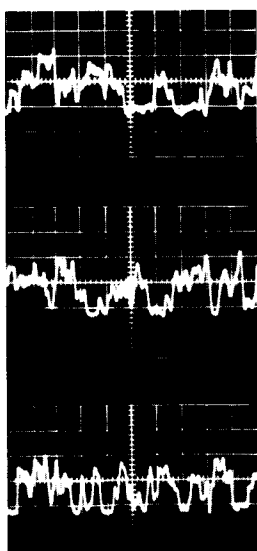


Static pressure
Traces 3 & 4, $x/d_o = 3.73$
Gain = 1.06 in. Hg/div.

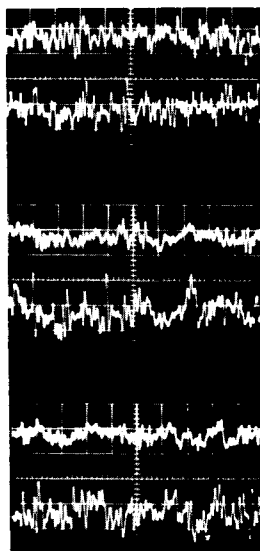


Total pressure
Trace 5, $x/d_o = 3.73$
Gain = 4.24 in. Hg/div.

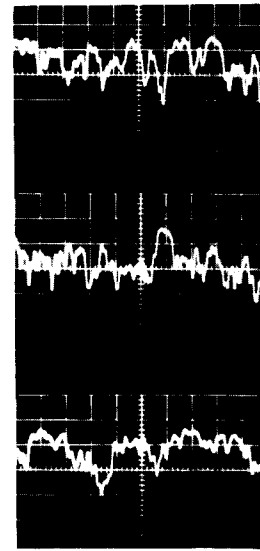
$$R = 1.84 \times 10^6$$



Static pressure
Trace 2, $x/d_o = 3.17$
Gain = 2.13 in. Hg/div.



Static pressure
Traces 3 & 4, $x/d_o = 3.73$
Gain = 2.13 in. Hg/div.

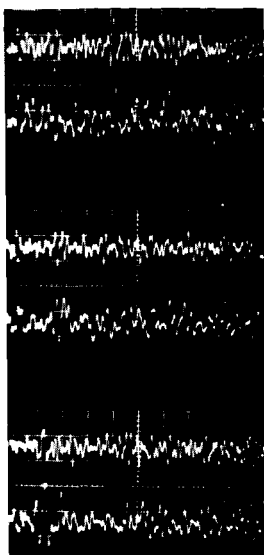


Total pressure
Trace 5, $x/d_o = 3.73$
Gain = 10.6 in. Hg/div.

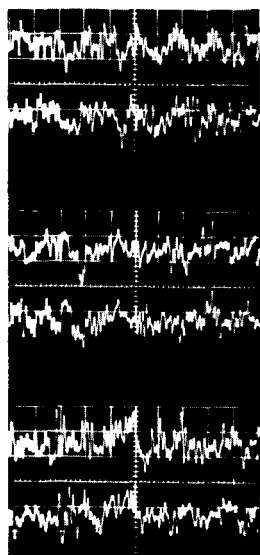
$$R = 2.63 \times 10^6$$

(a) Model 2-FC

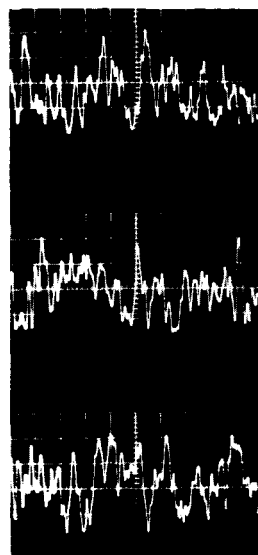
Figure 14.- Oscilloscope pictures of static- and total-pressure fluctuation (sweep rate equal 5 milliseconds per division).



Static pressure
Traces 1 & 2, $x/d_o = 3.17$
Gain = 0.625 in. Hg/div.

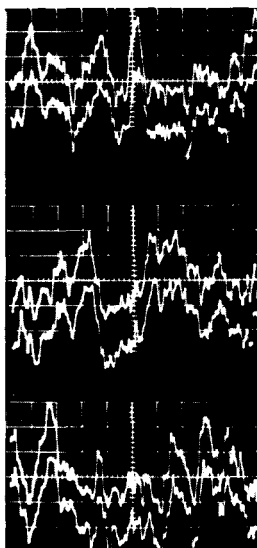


Static pressure
Traces 3 & 4, $x/d_o = 3.73$
Gain = 0.625 in. Hg/div.

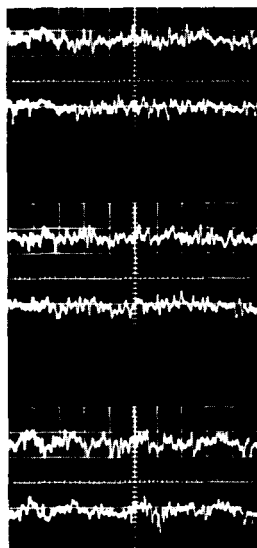


Total pressure
Trace 5, $x/d_o = 3.73$
Gain = 3.15 in. Hg/div.

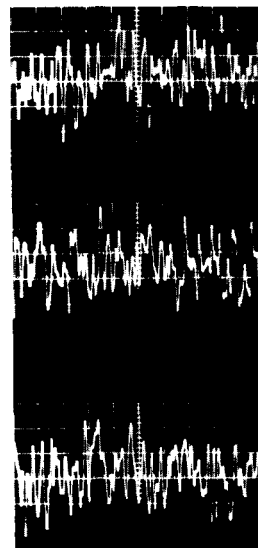
$$R = 1.84 \times 10^6$$



Static pressure
Traces 1 & 2, $x/d_o = 3.17$
Gain = 1.11 in. Hg/div.



Static pressure
Traces 3 & 4, $x/d_o = 3.73$
Gain = 1.11 in. Hg/div.

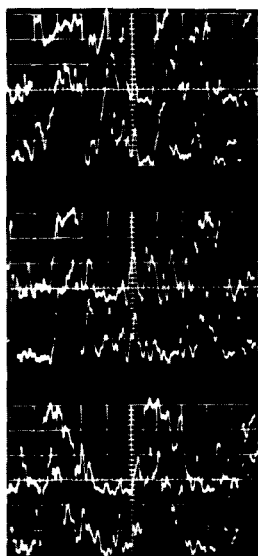


Total pressure
Trace 5, $x/d_o = 3.73$
Gain = 2.22 in. Hg/div.

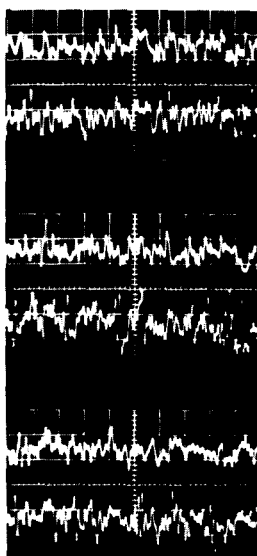
$$R = 2.63 \times 10^6$$

(b) Model 2-F

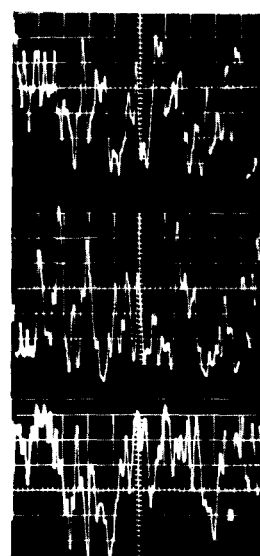
Figure 14.- Continued.



Static pressure
Traces 1 & 2, $x/d_o = 3.17$
Gain = 1.39 in. Hg/div.

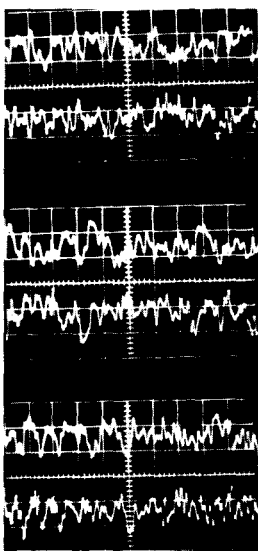


Static pressure
Traces 3 & 4, $x/d_o = 3.73$
Gain = 1.39 in. Hg/div.

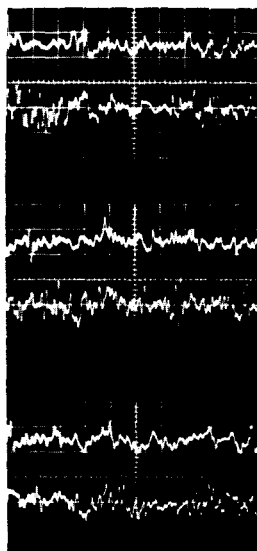


Total pressure
Trace 5, $x/d_o = 3.73$
Gain = 2.78 in. Hg/div.

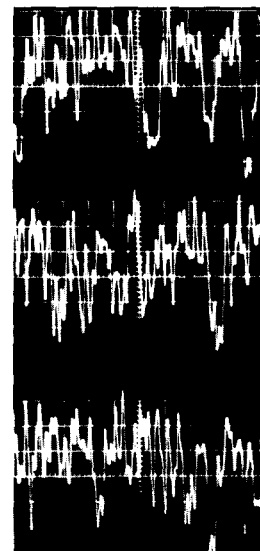
$R = 1.84 \times 10^6$



Static pressure
Traces 1 & 2, $x/d_o = 3.17$
Gain = 1.39 in. Hg/div.



Static pressure
Traces 3 & 4, $x/d_o = 3.73$
Gain = 1.39 in. Hg/div.

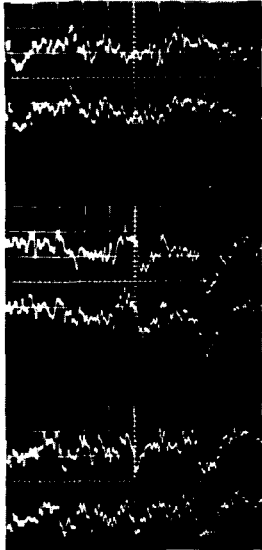


Total pressure
Trace 5, $x/d_o = 3.73$
Gain = 2.78 in. Hg/div.

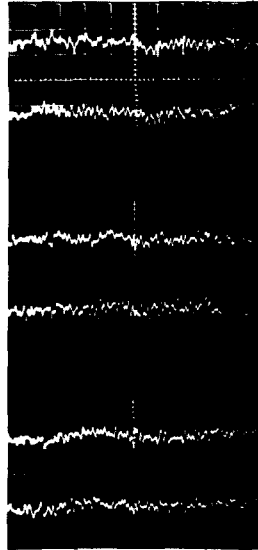
$R = 2.63 \times 10^6$

(c) Model 2-RC

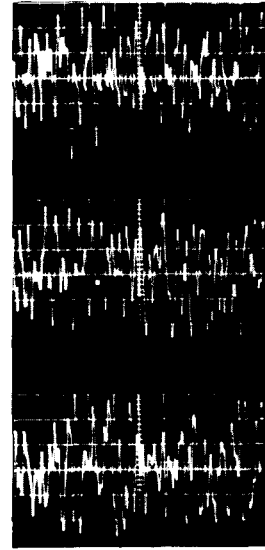
Figure 14.- Continued.



Static pressure
Traces 1 & 2, $x/d_o = 3.17$
Gain = 0.823 in. Hg/div.



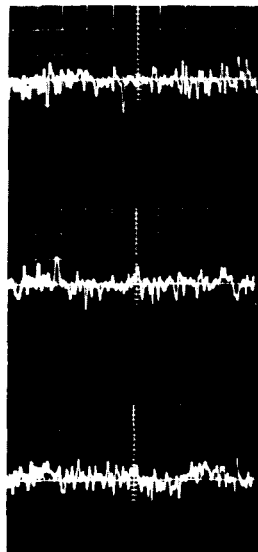
Static pressure
Traces 3 & 4, $x/d_o = 3.73$
Gain = 0.823 in. Hg/div.



Total pressure
Trace 5, $x/d_o = 3.73$
Gain = 0.823 in. Hg/div.

$$R = 2.63 \times 10^6$$

(d) Model 2-R



Static pressure
Trace 6, $x/d_o = 6.74$
Gain = 0.625 in. Hg/div.

$$R = 2.63 \times 10^6$$

(e) Model 2-F

Figure 14.- Concluded.

NOTES: (1) Reynolds number is based on the diameter of a circle with the same area as that of the capture area of the inlet.

(2) The symbol * denotes the occurrence of buzz.

Report and facility	Description			Test parameters				Test data				Performance		Remarks
	Configuration	Number of oblique shocks	Type of boundary-layer control	Free-stream Mach number	Reynolds number $\times 10^{-6}$	Angle of attack, deg	Angle of yaw, deg	Drag	Inlet-flow profile	Discharge-flow profile	Flow picture	Maximum total-pressure recovery	Mass-flow ratio	
TN D-854 Ames 1- by 3-Foot Supersonic Wind Tunnel		Isen-tropic	Scoop	3.70 to 3.82	1.84 to 2.63	0	0	No	No	Yes	No	0.898 at M = 3.8	1.0	Total and static-pressure fluctuation measurements in the throat region and static-pressure fluctuation measurements at the exit are presented.
TN D-854 Ames 1- by 3-Foot Supersonic Wind Tunnel		Isen-tropic	Scoop	3.70 to 3.82	1.84 to 2.63	0	0	No	No	Yes	No	0.898 at M = 3.8	1.0	Total and static-pressure fluctuation measurements in the throat region and static-pressure fluctuation measurements at the exit are presented.
TN D-854 Ames 1- by 3-Foot Supersonic Wind Tunnel		Isen-tropic	Scoop	3.70 to 3.82	1.84 to 2.63	0	0	No	No	Yes	No	0.898 at M = 3.8	1.0	Total and static-pressure fluctuation measurements in the throat region and static-pressure fluctuation measurements at the exit are presented.
TN D-854 Ames 1- by 3-Foot Supersonic Wind Tunnel		Isen-tropic	Scoop	3.70 to 3.82	1.84 to 2.63	0	0	No	No	Yes	No	0.898 at M = 3.8	1.0	Total and static-pressure fluctuation measurements in the throat region and static-pressure fluctuation measurements at the exit are presented.

Bibliography

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